NAVAL RESEARCH LABORATORY

NAVAL CENTER FOR SPACE TECHNOLOGY

Spacecraft Design Specification
for the
Full-sky Astrometric Mapping Explorer (FAME)

NCST-S-FM001 DRAFT **12 December 2000**

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Prepared in accordance with MIL-STD-961D

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RECORD OF CHANGES

REVISION LETTER	DATE	TITLE OR BRIEF DESCRIPTION	ENTERED BY
—	12 December 00	Draft Release for SRR	M. Ream

TABLE OF CONTENTS

LIST OF FIGURES

LIST OF TABLES

1. INTRODUCTION

1.1 Identification of Document

This is the Spacecraft Design Document for the Full-sky Astrometric Mapping Explorer (FAME) Observatory spacecraft bus. FAME is a NASA Medium Class Explorer (MIDEX) mission schedule for launch in 2004.

1.2 Scope of Document

This document applies to the FAME spacecraft bus being developed by the Naval Research Laboratory (NRL).

1.3 Purpose and Objectives of Document

The document establishes the performance, design, development, and test requirements for the FAME spacecraft bus.

1.4 Document Overview

This document is organized as follows:

Section 1, *Introduction:* The purpose and contents of this document, and an overview of the FAME program.

Section 2, *Referenced Documents:* A list of documents referenced in or required for use with this document.

Section 3, *Requirements:*

Section 4, *Qualification Provisions:*

Section 5, *Preparation For Delivery:*

Section 6, *Notes:*

2. RELATED DOCUMENTATION

The following documents of the exact issue shown form a part of this specification to the extent specified herein. In the event of conflict between the document referenced herein and the contents of this specification, the contents of this specification shall be considered a superseding requirement.

2.1 Parent Documents

The following documents are parent to this document:

Order S-13610-Y	Interagency Order funding FAME mission
NCST-D-FM001	Science Requirements Document for the Full-sky Astrometric Mapping Explorer (FAME)
NCST-D-FM002	Mission Requirements Document for the Full-sky Astrometric Mapping Explorer (FAME)
NCST-D-FM003	Project Management Plan for the for the Full-sky Astrometric Mapping Explorer (FAME)
NCST-D-FM004	Systems Engineering Management Plan (SEMP) for the Full-sky Astrometric Mapping Explorer (FAME)

2.2 Applicable Documents

The following documents are referenced within this volume or are directly applicable to this volume or contain polices or other directive matters that are binding on the content of this volume.

2.2.1 Government Documents

2.2.1.1 Specifications

MIL-M-38510J	Microcircuits, General Specification for
MIL-B-5087	

2.2.1.2 Standards

MIL-STD-1522	Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-1546	Parts, Materials, and Processes Control Program for Space and Launch Vehicles
MIL-STD-461E	Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment
MIL-STD-882D	System Safety
MIL-STD-975(M)	NASA Standard Electrical, Electronic, and Electromechanical (EEE) Parts List
MIL-STD-7179	Finishes, Coatings, and Sealants, for the Protection of Aerospace Weapons Systems
MSFC-SPEC-522B	Design Criteria for Controlling Stress Corrosion Cracking
NASA-STD-2100-91	NASA Software Documentation Standard (NSDS)

2.2.1.3 Handbooks

MIL-HDBK-217F	Reliability Prediction of Electronic Equipment
MIL-HDBK-1547A	Electronic Parts, Materials, and Processes for Space and Launch Vehicles
MIL-HDBK-1568	Material and Processes for Corrosion Prevention and Control in Aerospace Weapons Systems
NHB 1700.1B	NASA Safety Policy and Requirements Document

NHB 5300.4 (3A-2)	Soldering of Electrical Connections
NHB 5300.4 (3G)	Cable, Harness, and Wiring Interconnections
NHB 5300.4 (3H)	Crimping
NHB 5300.4 (3J)	Conformal Coating and Staking
NHB 5300.4 (3L)	ESD Control

2.2.1.4 Other Publications

EWRR 127-1	Eastern and Western Range Regulation 127-1, Range Safety Requirements
GSFC 311-INST-001	Instructions for EEE Parts Selection, Screening, and Qualification
GSFC-410-MIDEX-002	MIDEX Assurance Requirements (MAR)
NMI 7120.4	Management of Major System Program and Projects
NPG 7120.5A	NASA Program and Project Management Processes and Requirements
NPG 9501.2B	NASA Contractor Financial Management Reporting
NRP-1124	

Copies of specifications, standards, drawings, and publications required by suppliers in connection with specified procurement functions should be obtained from the contracting agency or as directed by the contracting officer.

2.2.2 Nongovernment Documents

ANSI/J-STD-001 through -006	High Reliability Soldering of Electrical Connections
EIA-625	Requirements for Handling Electrostatic Discharge Sensitive Devices
EIA-JEP-95	JEDEC Registered and Standard Outlines for Semiconductor Devices
IPC-C-406	Design and Application Guidelines for Surface Mount Connectors
IPC-D-275	Design Standard for Rigid Printed Boards and Rigid Printed Board Assemblies
IPC-D-279	Design Guidelines for Reliable Surface Mount Technology Printed Board Assemblies
IPC-D-322	Guidelines for Selecting Printed Wiring Board Sizes Using Standard Panel Sizes
IPC-D-325	Documentation Requirements for Printed Boards, Assemblies, and Support Drawings
IPC-D-330	Design Guide Manual
IPC-D-390	Automated Design Guidelines
IPC-EM-782	Surface Mount Design and Land Pattern Standard Spreadsheet
IPC-SM-782	Surface Mount Design and Land Pattern Standard
IPC-T-50	Terms and Definitions for Interconnecting and Packaging Electronic Circuits

Technical society and technical association specifications and standards are generally available for reference from libraries. They are also distributed among technical groups and using Federal Agencies.

2.3 Information Documents

The following documents, although not directly applicable, amplify or clarify the information presented in this volume.

3. REQUIREMENTS

3.1 Item Definition

The primary requirements of the spacecraft bus are to place the instrument in the proper orbit, provide a long-term stable platform for the instrument to collect science data, collect and forward the science data to the ground network, and de-orbit when the mission is complete. Requirements for each subsystem are provided in paragraphs 3.2.1.1 through 3.2.1.11.

3.1.1 Prime Item Diagrams

This paragraph shall incorproate, where applicable, either directly or by reference, the prime item level functional shcematics. This paragraph will cover the top-level functional flow diagrams of the configuration item and include digrammatic presentations to the level required to identify all essential functions.

The FAME spacecraft block diagram is shown in Figure 3-1. Block diagrams for each subsystem are provided in paragraphs 3.2.1.1 through 3.2.1.11.

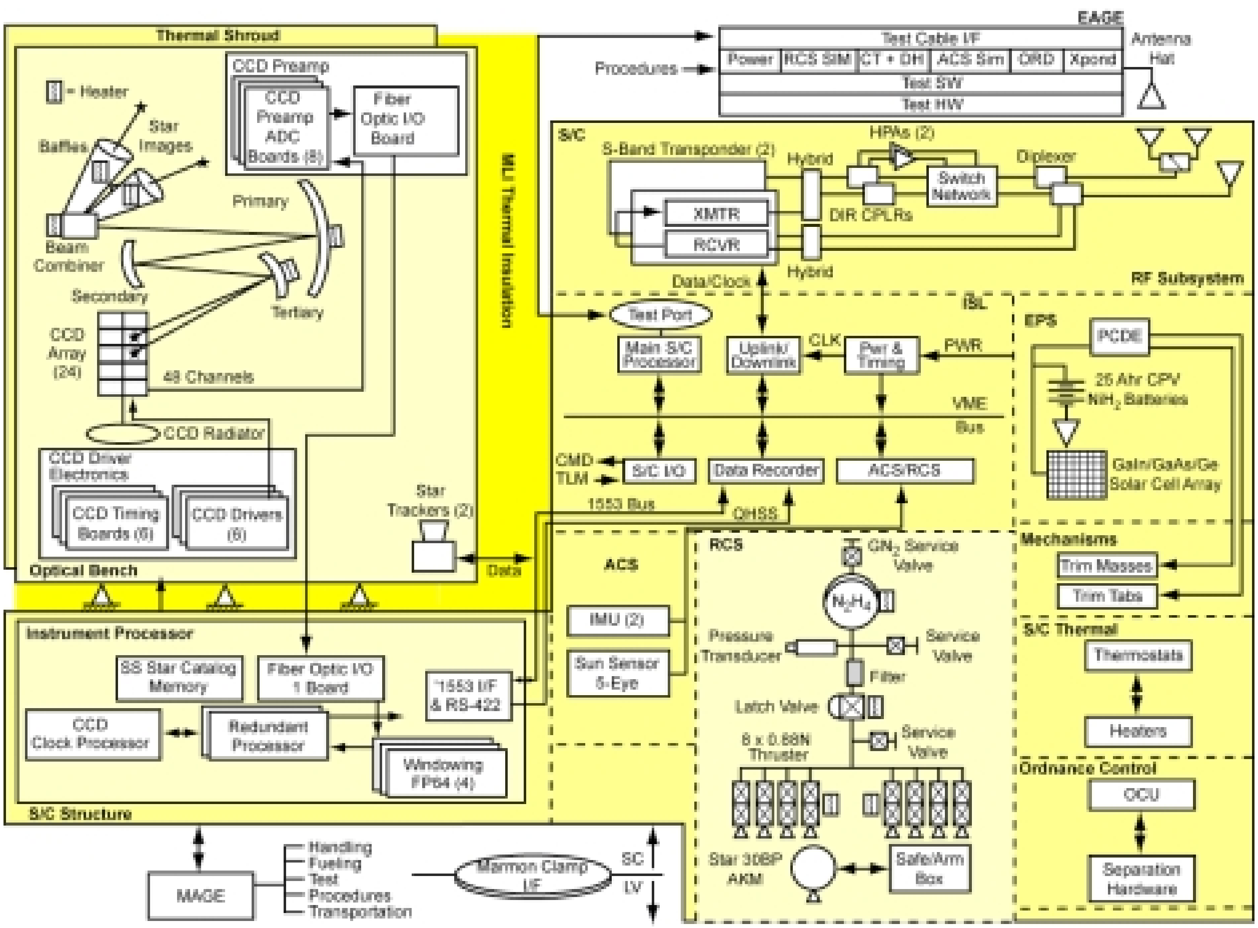

Figure 3-1. FAME Spacecraft Block Diagram

3.1.2 Interface Definition

Functional and physical interfaces between (a) this item and other configuraiton items, and (b) the major components within this this item. The functional interfaces shall be specified in quantitative terms of input/output voltages, accelerations, temperature ranges, shock limitations, loads, speeds, pitch and roll rates, etc. Where interfaces differ due to a change in operational mode, the requirements hsall be specified in a manner that identifies specific functional interface requirements for each different mode. Physical interface relationships shall be expressed in terms of dimensions with tolerances. This paragraph shall incorporate, either directly or by reference, interface control drawings, and other engineering data as necessary to define all functional and physical interfaces required to make the prime item compatible with other configuration items and to make components compatible with the prime item.

3.1.2.1 Spacecraft Bus to Instrument Mechanical/Thermal Interface

Mechanical interface

Volume interface

Mechanical environmental loads

20 ±2˚C conduction and TBD radiation thermal interface

TBD ±TBD˚C per minute thermal stability

Location, mechanical interface, and thermal interface for omnidirectional antenna

Location, mechanical interface, and thermal interface for star trackers

Alignment (instrument to spacecraft bus)

Allowable mass (instrument)

Jitter requirements

Environmental loads (manufacturing, transportation, testing, launch, on-orbit)

Location of spin balance masses for system testing or spin balance instrument before system integration

3.1.2.2 Spacecraft Bus to Instrument Electrical Interface

3.1.2.2.1 Power

a. Primary Power:

   (1) Voltage: Spacecraft bus power at 30 +TBD, -6 VDC, returns to spacecraft, single point ground (see Figure 3-2)

   (2) Power: TBD amperes per feed

   (3) Switching: Provided by spacecraft Power Control Distribution Electronics (PCDE)

   (4) Fault Protection: TBD provided by spacecraft PCDE

   (5) Inrush Current Limit: Limited to twice the average input operating current and shall settle to within 10% of nominal operating values within 200 milliseconds after application of power

b. Secondary Power: None.

c. Isolation: Primary input power and returns shall be isolated from the case (chassis) and secondary power circuitry by a minimum DC resistance of 1 megohm

d. Bonding/Grounding: Shall be in accordance with MIL-B-5087

e. Structure Grounding: Instrument structure shall be electrically grounded to the spacecraft bus with ground straps provided by NRL

f. Bonding: DC impedance shall be $\leq$2.5 mΩ for metal to metal surface, $\leq$10 Ω for metal to composite interfaces

g. EMI/EMC: MIL-STD-461 CS01, CS02, CS06, CE01, and CE03, RS (TBD per range requirements)

h. Instrument Power: Instrument power requirements are listed in Table 3-1

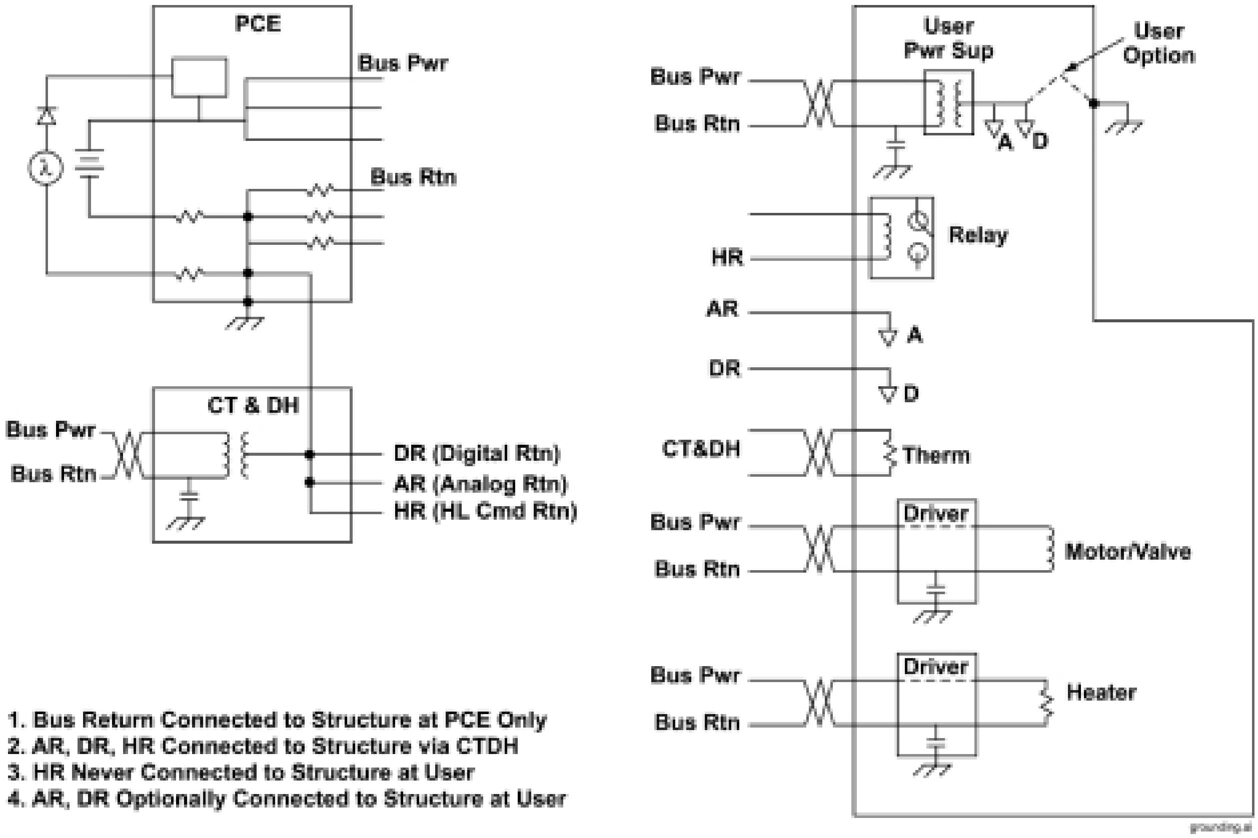

Figure 3-2. FAME Interface Grounding

Table 3-1. Instrument Power at 28 VDC

Component	Operational	Transfer Orbit	Survival
Focal Plane Assembly	6	0	0
Analog Processing Electronics	15	0	0
CCD Control Electronics	26	0	0
Data Processing and Instrument Control	50	0	0
Focal Plane Heaters	2	0	0
Instrument Heater	80	0	0
Survival Heaters	0	20	60
Total Power	179	20	60
Contingency	90	10	30
Design Limit	269	30	90

3.1.2.2.2 Command and Telemetry

The spacecraft and instrument shall use a redundant MIL-STD-1553 interface for command and most housekeeping data. The two redundant busses shall be controlled by the CT&DH subsystem of the spacecraft bus.

a. Command Interface:

(1) Time updates

(2) Star tracker attitude

(3) Active heater temperature setpoints

(4) Star catalog updates

(5) Flight software patches

(6) Special test modes

b. Housekeeping Data Interface:

(1) Temperatures

(2) Voltage/current monitors

(3) Command status information

(4) Attitude and spin rate

(5) Rate: 1 per second (WHAT RATE IS THIS? –Michael)

3.1.2.2.3 Analog Telemetry Interface

The spacecraft bus shall provide analog telemetry interfaces for temperature measurements (6 TBR) and voltage monitors (4 per side, 8 total TBR).

3.1.2.2.4 Mission Data Interface

The spacecraft bus shall accept science data from the instrument sent on a High Speed Serial (HSS) link (TBR) operating at rates from 0.5 to 20 megabits/second. The expected data rates are as follows:

a. Outside Galactic Plane: 817 stars/second = 263 kilobits/second

b. In Galactic Plane: Number of stars in the star catalog filtered to keep rate ≤400 kilobits/second

3.1.2.3 Spacecraft Bus to Launch Vehicle Interface

The spacecraft bus shall support a T0 interface with the launch vehicle and ground equipment.
The T0 interface shall provide a path for charging and maintenance of the battery.

The T0 interface shall provide a differential interface consisting of uplink clock and data and downlink clock and data to allow for ground control and monitoring of onboard subsystems.

The spacecraft bus shall provide a bi-level input to the launch vehicle to detect the separation of the upperstage from the spacecraft bus.

3.1.3 Major Components List

The FAME spacecraft bus consists of the following major subsystems:

a. Attitude Determination and Control System (ADCS) (paragraph 3.2.1.1)

b. Command, Telemetry, and Data Handling (CT&DH) Subsystem (paragraph 3.2.1.2)

c. Electrical Power Subsystem (EPS) (paragraph 3.2.1.3)

d. Flight Software (paragraph 3.2.1.4)

e. Harness Subsystem (paragraph 3.2.1.5)

f. Mechanisms Subsystem (paragraph 3.2.1.6)

g. Ordnance Subsystem (paragraph 3.2.1.7)

h. Radio Frequency Subsystem (RFS) (paragraph 3.2.1.8)

i. Reaction Control Subsystem (RCS) (paragraph 3.2.1.9)

j. Structures Subsystem (paragraph 3.2.1.10)

k. Thermal Control Subsystem (TCS) (paragraph 3.2.1.11)

3.1.4 Government Furnished Property List

Government services, facilities, or property are not required for the FAME mission. All government facility use is on a “full cost recovery” basis.

a. Launch Vehicle and Launch Services: A MedLite launch vehicle is provided by the NASA Launch Vehicle Office and is funded within the FAME investigation.

b. Deep Space Network: Use of the DSN is funded within the FAME investigation.

3.1.5 Government Loaned Property List

Not applicable.

3.2 Characteristics

3.2.1 Performance Characteristics

3.2.1.1 Attitude Determination and Control Subsystem (ADCS)

3.2.1.1.1 Spin Period

The spin period is as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.1.2 Spin Rate Variation

The spin rate variations shall be TBD, TBD, and TBD over 1.56 sec, 10 minutes, and 40 minutes time durations respectively during data collection phase of the mission.

3.2.1.1.3 Spin Rate Knowledge for Instrument

The spin rate knowledge for the instrument shall be 40 arcsec (1 sigma) for all three axes during data collection phase of the mission.

3.2.1.1.4 Spin Rate Knowledge for Bus

The spin rate knowledge for the instrument shall be TBD arcsec (1 sigma) for all three axes during data collection phase of the mission.

3.2.1.1.5 Sun Angle

The sun angle is as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.1.6 Sun Angle Variation

The sun angle variation shall be TBD degrees during data collection phase of the mission.

3.2.1.1.7 Sun Angle Knowledge

The sun angle knowledge shall be +- 0.5 degree during all phase of the mission.

3.2.1.1.8 Precession Period

The precession period is as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.1.9 Precession Rate Variation

The precession rate variations shall be TBD, TBD, and TBD over 1.56 sec, 10 minutes, and 40 minutes time durations respectively during data collection phase of the mission.

3.2.1.1.10 Precession Rate Knowledge

The precession rate knowledge shall be +- 60 arcsec during data collection phase of the mission.

3.2.1.1.11 Nutation Angle

The nutation angle about the TBD axis shall be +- 10 arcsec or less during data collection phase of the mission.

3.2.1.1.12 Nutation Angle Variation

The nutation angle variations shall be TBD, TBD, and TBD over 1.56 sec, 10 minutes, and 40 minutes time durations respectively during data collection phase of the mission.

3.2.1.1.13 Nutation Angle Knowledge

The nutation angle knowledge shall be +- TBD arcsec during data collection phase of the mission.

3.2.1.1.14 Inertial Attitude Knowledge

TBD

3.2.1.1.15 Transverse Rate Knowledge

TBD

3.2.1.1.16 Maximum Jitter

TBD

3.2.1.1.17 Mass Allocation

The maximum mass allocation for the ADCS subsystem shall not exceed TBD lbm.

3.2.1.1.18 Operational Modes

TBD

3.2.1.1.19 Data Collection

TBD

3.2.1.1.20 Inertial Pointing

TBD

3.2.1.1.21 Safe Hold

TBD

3.2.1.1.22 Open Loop Burn

TBD

3.2.1.1.23 Active Nutation Control

TBD

3.2.1.1.24 Spin Axis Precession

TBD

3.2.1.2 Command, Telemetry, and Data Handling Subsystem

A block diagram of the CT&DH subsystem is shown in Figure 3-3.

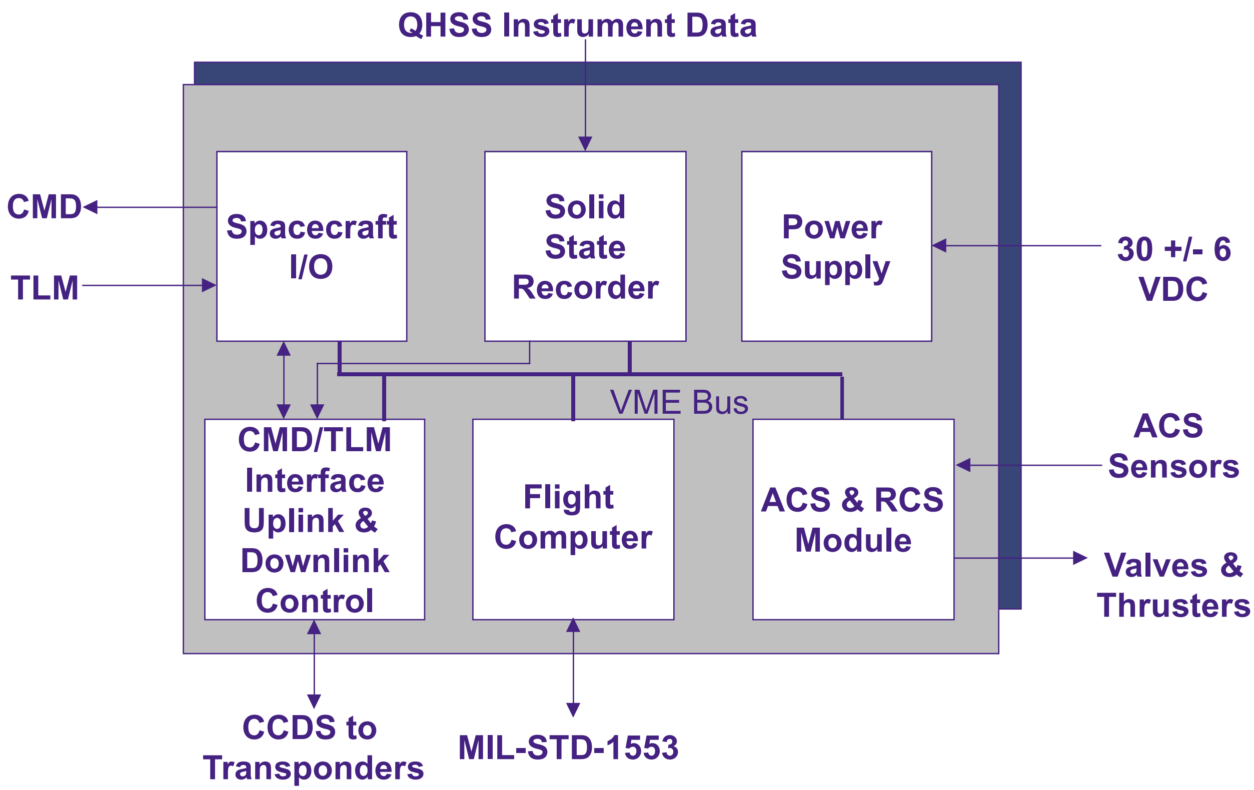

Figure 3-3. CT&DH Subsystem Block Diagram

3.2.1.2.1 Performance Requirements

The CT&DH subsystem shall communicate with FV and observatory subsystems (i.e., the ADCS, RCS, EPS, TCS, RFS, OCS, and instrument subsystems).

The CT&DH subsystem shall incorporate redundancy where practical to survive a Single Point Failure.

CT&DH functional requirements are listed in paragraphs 3.2.1.2.1.1 through 3.2.1.2.1.8

3.2.1.2.1.1 Uplink Deformatting and Decoding

The CT&DH subsystem shall be fully compatible with NASA Spaceflight Tracking Data Network (STDN) specifications as described in the FAME Detailed Mission Requirements Document, NCST-D-FM002.

No Communications Security (COMSEC) decryption shall be required.

The CT&DH subsystem shall decode, authenticate and process CCSDS commands received.

The CT&DH subsystem shall execute critical commands without CPU interaction.

The CT&DH subsystem shall transfer specific uplink data to instrument for control and reprogramming.

The CT&DH subsystem shall provide for an interface to Electrical Aerospace Ground Equipment (EAGE) for initial integration.

The CT&DH subsystem shall support the T-minus zero (T-0) interface to the ELV.

3.2.1.2.1.2 Downlink Formatting Formatting and Encoding

The CT&DH subsystem shall be fully compatible with NASA Spaceflight Tracking Data Network (STDN) specifications as described in the FAME Detailed Mission Requirements Document, NCST-D-FM002.

No Communications Security (COMSEC) encryption shall be required.

The CT&DH subsystem shall provide for downlink telemetry capability of 409.6 kbps of science and telemetry data.

The CT&DH subsystem shall provide error control coding and data interleaving to the downlink telemetry data stream.

3.2.1.2.1.3 Telemetry Gathering

The CT&DH subsystem shall collect SOH telemetry from all FV and observatory subsystems.

3.2.1.2.1.4 Command Processing

The CT&DH subsystem shall execute stored commands.

The CT&DH subsystem shall distribute commands to all FV and observatory subsystems.

3.2.1.2.1.5 Instrument Control

The CT&DH subsystem shall provide a power management capability.

The CT&DH subsystem shall provide the necessary commands to enable the observatory subsystem.

The CT&DH subsystem shall provide the necessary commands to open the protective outer doors of the observatory subsystem.

3.2.1.2.1.6 Attitude Determination and Control Processing and commanding

The CT&DH subsystem shall manage FV and observatory attitude for successful mission orbit insertion, stabilization, and de-orbit.

The CT&DH subsystem shall provide timers to achieve “safe distance” between third stage and FV prior to initiation of GTO insertion maneuvers.

3.2.1.2.1.7 Mission Data Buffering

The CT&DH subsystem shall collect, store, and buffer instrument payload science data generated at a maximum data rate of 4 Mbps for an average data rate of 320 kbps while using a maximum memory storage allocation of 4 Gigabits.

The CT&DH subsystem shall store instrument payload science and telemetry data up to the maximum allocated storage capability during scheduled or unscheduled downlink outages.

3.2.1.2.1.8 Time Distribution

The CT&DH subsystem shall maintain and distribute S/C bus time to all FV and observatory subsystems.

3.2.1.2.2 Interface Requirements

CT&DH subsystem interface requirements are listed in Table 3-2.

Table 3-2. CT&DH Subsystem Interfaces

Interface	Interface Type
Instrument interface	Quad High Speed Serial (QHSS) Mission Data
Attitude Determination and Control System (ADCS)	TBD Thrusters
	TBD Paraffin Actuators
	TBD Latch Valves
Spacecraft Interfaces	1553 Bus
	TBD Passive Analog Channels
	TBD Active Analog Channels
	TBD Bi-Levels
	TBD Hi-level Relay Drivers
	TBD Open Collector Drivers
Uplink/Downlink	2 kbps Uplink
	Up to 409.6 kbps Downlink

3.2.1.3 Electrical Power Subsystem

A block diagram of the EPS is shown in Figure 3-4. Performance requirements are listed in Table 3-3. Power requirements for major subsystems and components are listed in Table 3-4.

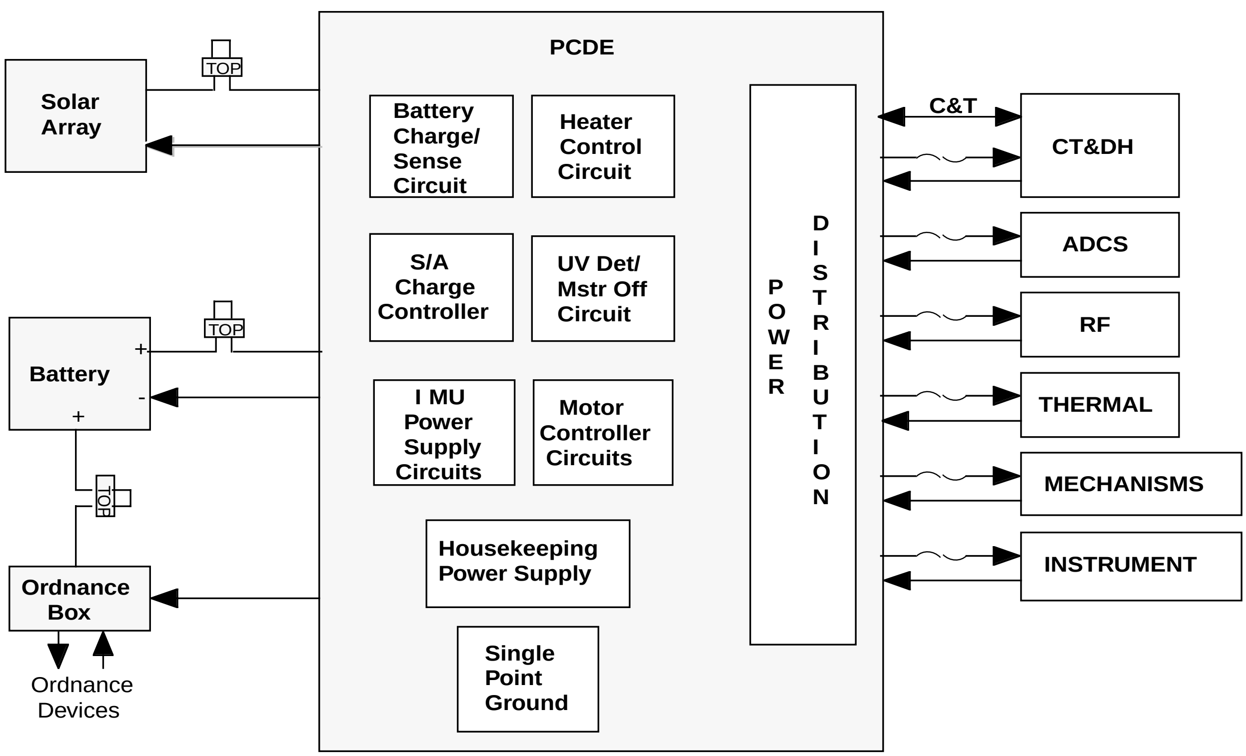

Figure 3-4. EPS Block Diagram

Table 3-3. EPS Requirements

Supply electrical power to all spacecraft subsystems during all mission phases
Provide sufficient solar array to support loads and battery charge
Provide energy storage to operate during seasonal eclipse periods, to support transient loads and to clear fault loads
Limit battery depth of discharge and overcharge; provide for system survival with undervoltage detection and non-critical load shed
Drive CG trim mass and radiation trim tab motors for precession control
Automatically reject excess solar array energy
Maintain spacecraft bus voltage at 30 +/- 6 Vdc
Design for autonomous operation to limit ground intervention; provide manual override of the autonomous functions
Install fault protection on non-critical loads and redundant critical loads
Distribute unswitched power to critical bus subsystems and switched power to all spacecraft subsystems
Monitor EPS health and provide EPS parameters to telemetry
Provide ordnance for satellite and interstage separation, solar array release and apogee kick motor
Ordnance design will meet safety requirements as specified in EWRR 127-1 for non-manned flights
Minimize magnetic moments
Design and fabricate spacecraft harness meeting requirements as specified in GSFC 311-INST-001
Design for 5 year mission; parts will handle temperature, radiation and mechanical extremes; parts will meet military Class B program requirements

Table 3-4. Power Requirements of Major Subsystems

Subsystem/Unit	Qty	Launch	Initial Acquisition/GTO	GEO/Ops	Safehold Mode
CT&DH	1	24.1	36.5	24.1	24.1
ADCS					
IMU	2	0	20	10	10
Sun Sensor and Electronics	1	1	1	1	1
Star Tracker	2	0	20	10	0
RF Subsystem					
Receiver	2	7.9	7.6	7.6	7.6
Transmitter	2	0	24	24	24
Power Amplifier	2	0	0	58	0
Mechanisms					
S/A Trim Tabs	6	0	0	0	0
Trim Mass Motors	6	0	0	0	0
EPS					
PCDU	1	15	15	15	15
Battery	1	0	0	0	0
S/C Heater Power		0	57.5	23.5	90.5
Instrument					
Electronics		0	0	99	0
Operational Heater		0	0	80	0
Survival Heater Feed		0	20	0	60
Subtotal by Operational Phase		47.7	201.6	352.2	232.2
(50% Instrument Margin)		0.0	10.0	89.5	30.0
(25% Design Margin)		11.9	45.4	43.3	43.1
Totals w/Margins		59.7	257.0	485.0	305.3

3.2.1.3.1 Bus Voltage

The Electrical Power Subsystem shall provide the S/C Bus electrical critical bus voltage of 30 +/- 6 VDC.

The Electrical Power Subsystem shall protect the S/C bus electrical critical bus from faults.

The Electrical Power Subsystem shall have the capability to power up from a low voltage condition.

3.2.1.3.2 Survival Power

The Electrical Power Subsystem shall provide stored electrical energy to the S/C Bus during mission eclipse periods.

3.2.1.3.3 Peak Power

3.2.1.3.4 Orbital Average Power

3.2.1.3.5 Battery Capacity

3.2.1.3.6 Power Distribution and Control

The Electrical Power Subsystem shall provide the capability to manage power distribution to the various subsystems of the spacecraft.

The Electrical Power Subsystem shall provide an unswitched critical power bus for utilization by the CT&DH and RF subsystems.

The Electrical Power Subsystem shall provide an unswitched critical power bus for utilization by the OCS subsystem.

3.2.1.3.7 Ordnance Control Box

3.2.1.3.8 Solar Array

3.2.1.3.9 Wire Harness

3.2.1.3.10 Spacecraft Dipole Moment

3.2.1.4 Flight Software

FAME flight software shall consist of the following Computer Software Configuration Items (CSCIs):

a. System Resource Manager (SRM) (paragraph 3.2.1.4.3.1): Provides boot processing, task management, memory management, timer services, inter-task communications, MIL-STD-1553 bus control, solid state recorder (SRR) management, diagnostic and logging services, and interrupt handling services for higher level processes.

b. Command and Telemetry (C&T) (paragraph 3.2.1.4.3.2): Decodes uplinked commands and data loads and distributes them to the appropriate software process or hardware entity. Gathers telemetry using an uploaded commutation format and distributes it to the onboard telemetry database, the onboard storage device, the Uplink/Downlink Module, or any combination thereof.

c. Stored Commanding (SC) (paragraph 3.2.1.4.3.3): An implementation of the commercial Spacecraft Command Language (SCL). Provides procedural scripts and the capability to respond autonomously via rule-based mechanisms to asynchronous events. Scripts and rules tasking the spacecraft are generated on the ground, compiled into tokens by SCL's compiler, formatted and uplinked, and stored onboard for subsequent execution. Script execution time can be absolute or relative.

d. Attitude Determination and Control (ADAC) (paragraph 3.2.1.4.3.4): Consists of the Guidance, Navigation, and Control (GNC) executive controlling spacecraft modes and allowable commands. ADAC generates commanded attitude and updates the spacecraft's inertial properties. ADCS passes data to and processes data from the Inertial Measurement Units (IMUs) and Star Tracker (STs). It manages spin rate, precession, and nutation during spin-stabilized flight; and controls thruster firing.

e. Orbit Propagation (OP) (paragraph 3.2.1.4.3.5): Accepts state vectors from the ground and propagates the spacecraft orbit over time. Delivers the current spacecraft position to ongoing software processes.

f. Payload Controller (PC) (paragraph 3.2.1.4.3.6):

g. Fault Detection Isolation and Recovery (FDIR) (paragraph 3.2.1.4.3.7):

FAME flight software shall control the following hardware modules:

a. Uplink/Downlink Module (UDM):

   (1) Uplink Task: Validates command formats and passes the commands on to the main CPU for execution or storage.

   (2) Downlink Task: Accepts a downlink stream from the main CPU, formats it, and supplies the data to the downlink at the selected rate.

b. Spacecraft I/O (SCIO):

c. ACS/RCS Module:

   (1) Thruster Management Task: Accepts specific commands from the main CPU and manages the thruster's hardware interface. It rejects invalid ADCS commands and returns an error code to the main CPU.

   (2) Sensor Management Task: Receives data from the IMU and the Sun Sensors and passes it on to the main CPU for processing.

d. Data Recorder and Interface Module (DRIM):

e. Power Converter and Time Distribution (PCOXO):

f. Flight Processor (FP):

3.2.1.4.1 Software Management Plan

FAME flight software shall be developed in accordance with NCST-SDP-FM001, *FAME Software Management Plan*.

3.2.1.4.2 Configuration Control

Configuration control of FAME flight software shall be in accordance with NCST-SDP-FM001, *FAME Software Management Plan*.

3.2.1.4.3 Software Functionality

Paragraphs 3.2.1.4.3.1 through 3.2.1.4.3.7 specify the software functionality of the CSCIs listed in paragraph 3.2.1.4.

3.2.1.4.3.1 System Resource Manager

3.2.1.4.3.2 Command and Telemetry

3.2.1.4.3.3 Stored Commanding

3.2.1.4.3.4 Attitude Determination and Control

3.2.1.4.3.5 Orbit Propagation

3.2.1.4.3.6 Payload Controller

3.2.1.4.3.7 Fault Detection and Recovery

3.2.1.5 Harness Subsystem

The harness subsystem shall comprise all FAME observatory system, subsystem, and component electrical interface cable assemblies.

a. The harness subsystem shall maintain the redundancy that is established within each individual component or assembly.

b. The harness subsystem shall be constructed of space-qualified wire, connectors, lacing, cord, shield, and other materials necessary to assure reliable electrical interconnections.

c. The harness subsystem shall be constructed to meet the EMC requirements as specified in paragraph 3.3.3.

d. The harness shall be designed to facilitate the removal and installation of the electronics assemblies as a unit.

e. To the extent practical, redundant wiring shall be routed through separate bundles and harness in-line interface connectors.

f. All safety critical functions shall be routed through separate connectors.

3.2.1.6 Mechanism Subsystem

3.2.1.6.1 Flight Vehicle Separation

3.2.1.6.1.1 Tipoff Rate

The Flight Vehicle tipoff rate shall be TBD deg/sec or less for the Flight Vehicle separation phase of the mission.

3.2.1.6.1.2 Kickoff Rate

The Flight Vehicle kickoff rate shall be TBD m/sec or less for the Flight Vehicle separation phase of the mission.

3.2.1.6.1.3 Redundancy

The Flight Vehicle separation shall provide redundant release devices or redundant initiators.

3.2.1.6.1.4 Shoe Ramp Angle

The Flight Vehicle separation shall have a shoe ramp angle of TBD degrees.

3.2.1.6.1.5 Separation Bolt Preload

The Flight Vehicle Separation bolts shall be preloaded to TBD.

3.2.1.6.1.6 Surface Compatibility

The separating surfaces shall be of compatible materials and have appropriate finishes or lubricants.

3.2.1.6.1.7 Ground handling V Clamp

The mechanism subsystem shall provide a ground handling V clamp for integration, test, and shipping phases of the mission.

3.2.1.6.2 Spacecraft Separation

3.2.1.6.2.1 Tipoff Rate

The spacecraft tipoff rate shall be TBD deg/sec or less for the Flight Vehicle separation phase of the mission.

3.2.1.6.2.2 Kickoff Rate

The spacecraft kickoff rate shall be TBD m/sec or less for the Flight Vehicle separation phase of the mission.

3.2.1.6.2.3 Redundancy

The spacecraft separation shall provide redundant release devices or redundant initiators.

3.2.1.6.2.4 Shoe Ramp Angle

The spacecraft separation shall have a shoe ramp angle of TBD degrees.

3.2.1.6.2.5 Separation Bolt Preload

The spacecraft Separation bolts shall be preloaded to TBD.

3.2.1.6.2.6 Surface Compatibility

The separating surfaces shall be of compatible materials and have appropriate finishes or lubricants.

3.2.1.6.2.7 Ground handling V Clamp

The mechanism subsystem shall provide a ground handling V clamp for integration, test, and shipping phases of the mission.

3.2.1.6.2.8 Orbital Debris

The mechanism subsystem shall retain all separation parts after spacecraft separation.

3.2.1.6.3 Solar Array Arm Deployment

3.2.1.6.3.1 Quantity and Locations

The mechanism subsystem shall provide a method of deploying size solar array assemblies positioned every 60 degrees around the spacecraft, one at each enclosure panel.

3.2.1.6.3.2 Panel Sweep Angle

After deployment, each solar array assembly shall be deployed to a final angle of TBD +-TBD degrees relative to the spacecraft geometric centerline.

3.2.1.6.3.3 Panel Radial Angle Error

After deployment, each solar array assembly shall be deployed to a final angle of TBD +-TBD degrees as defined by Figure X.X.

3.2.1.6.3.4 Panel Clocking Angle Error

After deployment, each solar array assembly shall be deployed to a final angle of TBD +-TBD degrees as defined by Figure X.X.

3.2.1.6.3.5 Web Sweep Angle

After deployment, each sun shade web shall be deployed to a final angle of TBD +-TBD degrees as defined by Figure X.X.

3.2.1.6.3.6 Web Radial Angle Error

After deployment, each sun shade web shall be deployed to a final angle of TBD +-TBD degrees as defined by Figure X.X.

3.2.1.6.3.7 Web Clocking Angle Error

After deployment, each sun shade web shall be deployed to a final angle of TBD +-TBD degrees as defined by Figure X.X.

3.2.1.6.3.8 Web Flatness

The sun shade flatness do to manufacturing shall be 2.5 mm over a 2 m span or less prior to system integration.

3.2.1.6.3.9 Redundancy

The solar array assemblies shall provide redundant release devices or redundant initiators.

3.2.1.6.3.10 Orbital Debris

The mechanism subsystem shall retain all deployment parts after solar array assembly deployment.

3.2.1.6.3.11 Stray Light

TBD

3.2.1.6.3.12 Minimum Frequency

TBD

3.2.1.6.4 Star Tracker Covers

3.2.1.6.4.1 Quantity

The mechanism subsystem shall provide two star tracker cover assemblies, one for each star tracker.

3.2.1.6.4.2 Contamination

The star tracker covers shall not allow the passage of a 0.001 inch diameter particle through the cover seal.

3.2.1.6.4.3 Total Life Cycles

The star tracker covers shall be able to operate TBD times during all phases of the mission.

3.2.1.6.4.4 Redundancy

TBD

3.2.1.6.4.5 Torque Margin

The star tracker covers shall have TBD torque margin during all phases of the mission.

3.2.1.6.4.6 Surface Compatibility

The sliding surfaces shall be of compatible materials and have appropriate finishes or lubricants.

3.2.1.6.4.7 Launch Retention

The star tracker covers shall have locking devices or sufficient torque margin to retain the covers during the launch and orbit insertion phases of the mission.

3.2.1.6.5 Trim Tabs

3.2.1.6.5.1 Quantity

The mechanism subsystem shall provide 6 trim tabs, one for each solar array assemblies positioned at the end for maximum control authority during all phases of the mission.

3.2.1.6.5.2 Degrees of Freedom

The trim tab shall have 1 degree of freedom, pitch up and down with respect to the solar array assembly.

3.2.1.6.5.3 Total Life Cycles

TBD

3.2.1.6.5.4 Redundancy

TBD

3.2.1.6.5.5 Torque Margin

The trim tabs shall have TBD torque margin during all phases of the mission.

3.2.1.6.5.6 Sweep Angle Error

TBD

3.2.1.6.5.7 Radial Angle Error

TBD

3.2.1.6.5.8 Clocking Angle Error

TBD

3.2.1.6.5.9 Tab Flatness

The trim tab flatness do to manufacturing shall be 1.25 mm over a 0.5 m span or less prior to system integration.

3.2.1.6.5.10 Angular Travel

The trim tab shall have an angular travel of +- 50 degrees with respect to the solar array panel at all phases of the mission.

3.2.1.6.5.11 Angular Resolution

The trim tab shall have a angular resolution of 0.02 degree at all phases of the mission.

3.2.1.6.5.12 Surface Area

Each trim tab shall have a surface area of TBD +- TBD m^2 at all phases of the mission.

3.2.1.6.5.13 Launch Retention

The trim tabs shall have locking devices or sufficient torque margin to retain the tabs during the launch and orbit insertion phases of the mission.

3.2.1.6.6 Trim Masses

3.2.1.6.6.1 Quantity

The mechanism subsystem shall provide 6 trim masses, 4 for CG control and 2 for moments of inertia control for all phases of the mission.

3.2.1.6.6.2 Mass

The mechanism subsystem shall provide 6.0 kg mass for the 6 CG control trim masses and 3.3 kg mass for the 2 moments of inertia trim masses.

3.2.1.6.6.3 Linear Travel

The mechanism subsystem shall provide a travel of +- 0.5 m for the 6 CG control trim masses and a travel of +- 0.35 m for the 2 moments of inertia trim masses.

3.2.1.6.6.4 Linear Resolution

TBD

3.2.1.6.6.5 Location Knowledge

TBD

3.2.1.6.6.6 Transverse Variation

TBD

3.2.1.6.6.7 Launch Retention

The trim mass shall have locking devices or sufficient torque margin to retain the masses during the launch and orbit insertion phases of the mission.

3.2.1.6.6.8 Total Life Cycles

TBD

3.2.1.6.6.9 Redundancy

TBD

3.2.1.6.6.10 Torque Margin

The trim masses shall have TBD torque margin during all phases of the mission.

3.2.1.6.6.11 Alignment Reference

TBD

3.2.1.6.7 Mass Allocation

The maximum mass allocation for the mechanism subsystem shall not exceed TBD lbm.

3.2.1.6.8 Power

TBD

3.2.1.6.9 Bus Voltage

TBD

3.2.1.6.10 Maximum Jitter

TBD

3.2.1.7 Ordnance Control Subsystem (OCS)

The OCS shall provide the necessary switching and control logic to enable, arm, and execute S/C bus ordnance functions according to the requirements of MIL-STD-1576.

a. The ordnance shall provide the following functions: (i) spacecraft and AKM separation; (ii) AKM inhibit, arm and fire; (iii) solar panel deployments; and (iii) RCS enable.

b. The OCS electronic circuitry shall be configured to provide the necessary interrupt and fault tolerance.

c. The OCS shall be analyzed as specified in paragraph 3.2.3 to ensure no Single Point Failures (SPFs) exist and that the system is single fault tolerant to ensure mission success.

d. The OCS shall operate in concert with the other S/C bus subsystems and the EPS.

e. [High current dc power shall be derived from the battery via a special ordnance-only bus.]

f. The OCS shall output pulsed high-current energy to the electro-explosive devices (EEDs) via ordnance safe and arm connectors.

g. The ordnance safe and arm connectors shall be located in the outboard wiring harnesses.

h. The NASA Standard Initiator (NSI) shall be used in the OCS.

3.2.1.8 Radio Frequency Subsystem

Provide spacecraft 2 kbps command capability at any vehicle orientation

Provide low rate 1 kbps minimum telemetry for emergency, safehold, and initial acquisition operations at any vehicle orientation

Provide high rate data downlink at 409 kbps

Provide coherent spacecraft range and range rate capability

Compatible with NASA Space Tracking Data Network (STDN) and NRL ground stations

Consultative Committee on Space Data Systems (CCSDS) compatible

Comply with National Telecommunications Information Administration (NTIA) frequency management regulations

3.2.1.8.1 Uplink

S-band uplink (2025 to 2120 MHz)

Command data = 2 kbps, NRZ-M data BPSK modulated synchronously on a 16 kHz sinewave subcarrier

Uplink subcarrier modulation index = 1 radian peak

Output to CTDH

Table 3-5. Uplink Interface to CT&DH Subsystem

Data	RS-422	NRZ-L
Clock	RS-422	Rising edge of clock at midpoint of data bit
Subcarrier lock	TTL	
Receiver lock	TTL	

Table 3-6. Uplink Budget (Omni Antenna, Geosynchronous Orbit)

Transmitter power (200 W)	53.0 dBm
Line and diplexer loss	-2.0 dB
Antenna gain (10 m)	44.0 dBi
Free space loss (geosynchronous at 5 degrees elevation)	-190.3 dB
Minimum antenna gain	-15.0 dBi (includes hybrid loss)
-Receiver sensitivity	-(-118.0 dBm)
Margin	7.7 dB

3.2.1.8.2 Downlink

S-band downlink 2200 to 2300 MHz (turnaround ratio 240/221)

Low data rate:

1 kbps (2.29 ksps) NRZ-M, bpsk modulated onto 1.7 MHz subcarrier

Phase modulated onto S-band carrier at modulation index of 1.6 radians peak

High data rate: 409 kbps (936.8 ksps) NRZ-M, BPSK modulated onto S-band carrier

Input from CTDH (same for both low and high data rate

Table 3-7

Data	RS-422	NRZ-L
Clock	RS-422	Rising edge of clock at midpoint of data bit

Table 3-8. Downlink Budget (409 kbps, Geosynchronous Orbit)

Transmitter power (w/SSPA)	40.0 dBm (10 W)
Diplexer and switch loss	-1.5 dB
Line loss	-2.0 dB
Antenna gain	-1.0 dBi
Free space loss (5 degrees elevation)	-191.8 dB
Atmosphere loss (5 degrees elevation)	-0.5 dB
Data rate	-56.1 dB Hz
Receive G/T	22.3 dB/K
Boltzmann's constant	198.6 dBm/Hz/K
Eb/No	8.0 dB
Implementation loss	-2.0 dB
Required Eb/No (10^{-6} Bit Error Rate)	-3.0 dB
Margin	3.0 dB

Table 3-9. Downlink Budget (1 kbps, Geosynchronous Orbit)

Transmitter power	33.0 dBm (2 W)
Modulation loss	-2.3 dB
Diplexer and switch loss	-1.5 dB
Line loss	-2.0 dB
Antenna gain	-15.0 dBi (includes hybrid loss)
Free space loss (5 degrees elevation)	-191.8 dB
Atmosphere loss (5 degrees elevation)	-0.5 dB
Data rate	-30.0 dB Hz
Receive G/T	22.3 dB/K
Boltzmann's constant	198.6 dBm/Hz/K
Eb/No	10.8 dB
Implementation loss	-2.0 dB
Required Eb/No (10^{-6} Bit Error Rate)	-3.0 dB
Margin	5.8 dB

3.2.1.8.3 Ranging

Ranging signal phase modulated directly onto uplink carrier at modulation index of 0.5 radian

Sequential square wave ranging (1.01 kHz to 515 kHz)

Ranging signal demodulated from uplink and phase modulated directly onto downlink carrier at modulation index of 0.5 radian

Downlink carrier reference generated from uplink carrier for coherent operation

Non-coherent downlink operation using on-board reference oscillator when uplink is not present

3.2.1.8.4 Electrical Input/Output

a. Transponder Commands: 26 V pulses to latching relays

b. Transponder Telemetry: Discrete 5 V, 0 to 5 V analog, and passive analog for temperature monitoring

c. Uplink Command Output:

    (1) Clock and data: RS-422, NRZ-L

    (2) Receiver lock and subcarrier lock: TTL

d. Downlink Data Input: Clock and data, RS-422, NRZ-L

e. Solid State Power Amplifier (SSPA) On/Off: 26 V pulses to latching relay

f. Transfer switches and Single Pole Double Throw (SPDT) RF Switches: 26 V pulses to latching relays

g. DC Power:

    (1) XMTR: 24 W at 22 to 36 V

    (2) RCVR: 4 W at 22 to 36 V

    (3) SSPA: 63 W at 22 to 36 V

3.2.1.9 Reaction Control Subsystem

3.2.1.9.1 Three Axis Attitude Control

The on-board propulsion systems shall provide 3-axis attitude control for all Flight Vehicle and spacecraft mission phases including pointing slew maneuvers, spin-up, spin axis precession (SAP), active nutation control (ANC), spin down, and attitude control during the on-board delta velocity maneuvers. The propulsion system will perform spin axis ANC during the spin stabilized SRM firing rather than full three axis attitude control.

3.2.1.9.2 Spacecraft Delta Velocity

The on-board propulsion system shall provide the vehicle total delta velocity between the SRM drop-off orbit, GEO mission orbit, and final spacecraft disposal orbit of 22 m/sec.

3.2.1.9.3 Safety Factors

TBD

3.2.1.9.4 Mechanical Activation

TBD

3.2.1.9.5 Electrical Activation

TBD

3.2.1.9.6 Surge Pressure

TBD

3.2.1.9.7 Mass Allocation

The maximum mass allocation for the RCS subsystem shall not exceed TBD lbm.

3.2.1.9.8 Component Cleanliness

The RCS subsystem shall follow the cleanliness guidelines as stated in MIL-STD-1246 100A.

3.2.1.9.9 Power

TBD

3.2.1.9.10 Bus Voltage

TBD

3.2.1.9.11 External Leakage

TBD

3.2.1.9.12 Range Safety Verification Support

TBD

3.2.1.9.13 Valve Leakage

TBD

3.2.1.9.14 Alignment Verification Plan

TBD

3.2.1.9.15 Thrusters

3.2.1.9.15.1 Thruster Control Authority Margin

The propulsion subsystem shall provide a system with a thruster control authority margin of 25 % during all phases of the mission.

3.2.1.9.15.2 Simultaneous Thruster Firings

TBD

3.2.1.9.15.3 Thruster Pulses

The thrusters shall be capable of pulsing 200,000 times during the life of the mission.

3.2.1.9.15.4 Spacecraft Impulse Bit

The thrusters shall have a minimum impulse bit for the spacecraft of .013 N*sec during the life of the mission.

3.2.1.9.15.5 Flight Vehicle Impulse Bit

The thrusters shall have a minimum impulse bit for the Flight Vehicle of TBD N*sec during the life of the mission.

3.2.1.9.15.6 Spacecraft Thrust Levels

The thrusters shall have a thrust level of 0.19 to TBD N for the spacecraft during the life of the mission.

3.2.1.9.15.7 Flight Vehicle Thrust Levels

The thrusters shall have a thrust level of TBD to TBD N for the Flight Vehicle during the life of the mission.

3.2.1.9.15.8 Spacecraft Burn Duration

The thrusters shall have a burn duration of 10 to TBD msec for the spacecraft during the life of the mission.

3.2.1.9.15.9 Flight Vehicle Burn Duration

The thrusters shall have a burn duration of TBD to TBD msec for the Flight Vehicle during the life of the mission.

3.2.1.9.15.10 Cold Starts

TBD

3.2.1.9.15.11 Total Impulse per Thruster

The thrusters shall have a total impulse of 80,000 N*sec.

3.2.1.9.15.12 Mechanical Alignment

TBD

3.2.1.9.15.13 Mechanical Alignment Knowledge

TBD

3.2.1.9.15.14 Protective Covers

TBD

3.2.1.9.16 Propellant Hardware

3.2.1.9.16.1 Tank Design

The propellant tank shall be design to MIL-STD-1522.

3.2.1.9.16.2 Propellant Center of Mass

TBD

3.2.1.9.16.3 Propellant Slosh

TBD

3.2.1.9.16.4 Propellant Slosh Frequency

TBD

3.2.1.9.16.5 Ground Support Equipment

The RCS subsystem shall provide the necessary ground support equipment for the integration, test, and field operation phases of the mission.

3.2.1.9.16.6 Fill Valve Protective Covers

TBD

3.2.1.9.16.7 Tank Protective Covers

3.2.1.9.17 Propellant

3.2.1.9.17.1 Type

TBD

3.2.1.9.17.2 Mass Allocation

TBD

3.2.1.9.17.3 Servicing Equipment

TBD

3.2.1.9.18 Apogee Kick Motor (AKM)

3.2.1.9.18.1 Delta Velocity

The AKM shall have a Delta Velocity of 1478 m/sec or greater prior to orbit insertion phases of the mission.

3.2.1.9.18.2 Mass Allocation

The maximum mass allocation for the RCS subsystem shall not exceed TBD kg.

3.2.1.9.18.3 Case Leakage

TBD

3.2.1.9.18.4 Acceleration

The AKM shall provide an acceleration of TBD g's for the Flight Vehicle for the orbit insertion phase of the mission.

3.2.1.9.18.5 Offload

The AKM shall be offload 20 % of its propellant by the AKM vendor for the orbit insertion phases of the mission.

3.2.1.9.18.6 Spin Rate

The AKM shall be design to operate are a spin rate of 60 RPM prior to orbit insertion phase of the mission.

3.2.1.9.18.7 Static Balance

The CG of the AKM shall be TBD or less inch lateral offset from the thrust centerline during the orbit insertion phases of the mission.

3.2.1.9.18.8 Dynamic Balance

The principle spin axis of the AKM shall be TBD or less degrees offset from the AKM thrust centerline during the orbit insertion phases of the mission.

3.2.1.9.18.9 Mass Simulator

The RCS subsystem shall provide a AKM mass simulator for system level testing phase of the mission.

3.2.1.9.18.10 Thermal Simulator

TBD

3.2.1.9.18.11 Thermal Model

The AKM vendor shall provide a verified thermal model of the AKM for system level thermal analysis.

3.2.1.9.18.12 Shipping Container

The AKM vendor shall provide a AKM shipping container for system level integration and the launch facility.

3.2.1.9.18.13 Ground Support Equipment

TBD

3.2.1.9.18.14 Interface Verification Plan

TBD

3.2.1.10 Structures Subsystem

3.2.1.10.1 Mass Allocation

The maximum mass allocation for the Flight Vehicle with all separation hardware shall not exceed 1100 kg.

3.2.1.10.2 Stowed Envelope

The stowed envelope is as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.10.3 Deployed Configuration

TBD

3.2.1.10.4 Coordinate System

The coordinate system is as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.10.5 Solar Array Panel Area

The solar cell area for each solar array panel shall be TBD inches or greater in the stowed configuration.

3.2.1.10.6 Solar Array Panel Flatness

The solar array panel flatness do to manufacturing shall be 2.5 mm over a 2 m span or less prior to system integration.

3.2.1.10.7 Sun Shade Web Flatness

The sun shade web flatness do to manufacturing shall be 2.5 mm over a 2 m span or less prior to system integration.

3.2.1.10.8 Electronics Deck Flatness

The electronics deck flatness do to manufacturing shall be 2.5 mm over a 2 m span or less prior to system integration.

3.2.1.10.9 Electronics Deck Radiating Surface

The electronics deck shall have a surface area of TBD inches to provide enough surface for radiating ....TBD

3.2.1.10.10 Locations

3.2.1.10.10.1 Sun Sensor

The sun sensor shall be positioned 90 degrees from the spin axis with an unobstructed field of view during all phases of the mission.

3.2.1.10.10.2 Star Tracker

The star trackers shall be positioned , if possible, 90 degrees from the spin axis with an unobstructed field of view during all phases of the mission.

3.2.1.10.10.3 IMU

The IMU shall be positioned such that the IMU reference surface maybe accessible during system integration.

3.2.1.10.10.4 Trim Tab

The trim tabs shall be positioned at the end of each solar array panel to provide maximum control authority.

3.2.1.10.10.5 Trim Mass

The trim masses shall be positioned as defined by the ADCS subsystem to provide maximum control authority for both Center of Mass (CG) and principle moments of inertia for the spin axis control.

3.2.1.10.10.6 Propellant Tank

The propellant tank shall be positioned such that the depletion of propellant will have minimum effect on the spacecraft CG location during all phases of the mission.

3.2.1.10.10.7 Thrusters

The thrusters shall be positioned as defined by the RCS subsystem to provide maximum control authority for body motion as defined by the ADCS subsystem.

3.2.1.10.10.8 Propellant Fill and Drain Valves

The fill and drain valves shall be easily accessible during the integration, test, and field operation phases of the mission.

3.2.1.10.10.9 Apogee Kick Motor (AKM)

The AKM shall be positioned as defined by the RCS subsystem with an unobstructed path for the plume to provide maximum control authority for orbit insertion phases of the mission.

3.2.1.10.10.10 RF Antennas

The RF antennas shall be positioned as defined by the RF subsystem to provide maximum link margin for all phases of the mission.

3.2.1.10.10.11 Instrument

The instrument shall be positioned at the +Z location of the spacecraft bus.

3.2.1.10.10.12 Instrument Orientation

The instrument shall be oriented such that the entrance apertures look 90 +- TBD degrees from the spacecraft spin axis during data collection phase of the mission.

3.2.1.10.10.13 Ordnance Arming Plugs

The ordnance arming plugs shall be easily accessible during the integration, test, and field operation phases of the mission.

3.2.1.10.10.14 Separation Connectors

The separation connectors shall be easily accessible during the integration, test, and field operation phases of the mission.

3.2.1.10.11 Mass Properties

3.2.1.10.11.1 FV Static Balance

The CG of the Flight Vehicle (FV) shall be TBD or less inch lateral offset from the FV geometric centerline during the orbit insertion phases of the mission.

3.2.1.10.11.2 FV Dynamic Balance

The principle spin axis of the Flight Vehicle (FV) shall be TBD or less degrees offset from the FV geometric centerline during the orbit insertion phases of the mission.

3.2.1.10.11.3 SC Static Balance

The CG of the spacecraft shall be 0.39 or less inch lateral offset from the spacecraft geometric centerline during the start of data collection phase of the mission.

3.2.1.10.11.4 SC Axial CG Location

The CG of the spacecraft shall be 31.5 +-4.0 inches axial distance from the solar array panel during data collection phase of the mission.

3.2.1.10.11.5 SC Principle Spin Axis

The principle spin axis of the spacecraft shall be 1.5 or less degrees offset from the spacecraft geometric centerline during the start of data collection phase of the mission.

3.2.1.10.11.6 SC Principle Spin Axis Inertia

The spin moment of inertia of the spacecraft shall be 1,298,000 to 1,435,000 lbm*in^2 during the data collection phase of the mission.

3.2.1.10.11.7 SC Principle Transverse Axes Inertia

Both transverse moment of inertias of the spacecraft shall be 89 to 91% of the principle spin axis during the data collection phase of the mission.

3.2.1.10.11.8 SC Transverse Product of Inertia

The transverse product of inertia of the spacecraft shall be 13,700 lbm*in^2 or less during the data collection phase of the mission.

3.2.1.10.11.9 Mass Reports

The mass reports are as stated in the FAME Mission Requirements Document, NCST-D-FM002.

3.2.1.10.12 Maximum Jitter

TBD

3.2.1.10.13 Design Limit Loads (DLL)

The DLL are as stated in the FAME Design, Loads, and Analysis Plan, NCST-D-FM017.

3.2.1.10.14 Natural Frequency

The natural frequencies are as stated in the FAME Design, Loads, and Analysis Plan, NCST-D-FM017.

3.2.1.10.15 Factors of Safety (FOS)

The FOS are as stated in the FAME Design, Loads, and Analysis Plan, NCST-D-FM017.

3.2.1.10.16 FV Lifting Points

The FAME spacecraft design shall incorporate provisions for lifting the fully assembled flight vehicle. These provisions shall be assessable at the flight vehicle level of assembly and shall be capable of safely supporting the fully assembled flight vehicle mass.

3.2.1.10.17 SC Lifting Points

The FAME spacecraft design shall incorporate provisions for lifting the bus with and without the instrument or instrument mass simulator.

3.2.1.10.18 Interstage Lifting Points

The interstage design shall incorporate provisions for lifting the insterstage and the interstage/SRM assembly.

3.2.1.10.19 SC Shipping Container

An appropriately sized and designed shipping container shall be used for transportation of the FAME spacecraft to the launch site and any non-NRL testing facilities. The FAME program has arranged with the GSFC Explorers Office for use of the XTE/TRMM shipping container for this purpose.

3.2.1.10.20 FV Shipping Container

An appropriately sized and designed shipping container shall be used for transportation of the FAME Flight Vehicle at the launch site as required.

3.2.1.10.21 Mass Simulators

Provide all mass simulators necessary to support bus, instrument, and spacecraft testing.

3.2.1.10.22 Alignment Reference

The bus design shall incorporate provisions for mounting optical reference surfaces as required to support alignment activities.

3.2.1.10.23 Test Fixtures

All necessary test fixtures shall be provided to support component and system level testing. Test fixtures shall be designed with appropriate factors of safety.

3.2.1.10.24 Handling Fixtures

All necessary handling and lift fixtures shall be provided to support integration and test, transportation and field operations. Fixtures shall be designed with appropriate factors of safety and tested in accordance with NRL and range safety requirements.

3.2.1.10.25 Assembly Fixtures

Provide all necessary tooling and assembly fixtures for efficient and accurate fabrication and assembly of FAME flight hardware.

3.2.1.10.26 Deployment Fixtures

Deployment test fixtures shall be provided to support development and system level testing. Fixtures shall be designed to prevent damage to flight hardware during development and acceptance testing.

3.2.1.11 Thermal Control Subsystem

3.2.1.11.1 Temperature

3.2.1.11.1.1 Non-Operational

3.2.1.11.1.1.1 Electronics Boxes

The TCS subsystem shall provide a non-operational interface temperature of –10 to 50 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.2 Battery

The TCS subsystem shall provide a non-operational interface temperature of 0 to 30 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.3 Thruster Valves

The TCS subsystem shall provide a non-operational interface temperature of 5 to 40 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.4 Propellant

The TCS subsystem shall provide a non-operational interface temperature of 5 to 40 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.5 AKM Case

The TCS subsystem shall provide a non-operational interface temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.6 AKM Nozzle

The TCS subsystem shall provide a non-operational interface temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.7 Instrument Interface

The TCS subsystem shall provide a non-operational interface temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.8 Star Tracker

The TCS subsystem shall provide a non-operational interface temperature of –30 to 50 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.9 Omni Antenna

The TCS subsystem shall provide a non-operational interface temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.10 Trim Motor

The TCS subsystem shall provide a non-operational interface temperature of –40 to 80 degrees C for the non-operational phases of the mission.

3.2.1.11.1.1.11 Release Devices

The TCS subsystem shall provide a non-operational interface temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.2 Operational

3.2.1.11.1.2.1 Electronics Boxes

The TCS subsystem shall provide an operational interface temperature of 0 to 40 degrees C for the operational phases of the mission.

3.2.1.11.1.2.2 Battery

The TCS subsystem shall provide an operational interface temperature of 0 to 30 degrees C for the operational phases of the mission.

3.2.1.11.1.2.3 Thruster Valves

The TCS subsystem shall provide an operational interface temperature of 5 to 30 degrees C for the operational phases of the mission.

3.2.1.11.1.2.4 Propellant

The TCS subsystem shall provide an operational interface temperature of TBD to TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.5 AKM Case

The TCS subsystem shall provide an operational case temperature of TBD to TBD degrees C for the non-operational phases of the mission.

3.2.1.11.1.2.6 AKM Case Gradients

The TCS subsystem shall provide an operational case temperature gradient of TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.7 AKM Nozzle

The TCS subsystem shall provide an operational nozzle temperature of TBD to TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.8 Instrument Interface

The TCS subsystem shall provide an operational interface temperature of 18 to 22 degrees C for the operational phases of the mission.

3.2.1.11.1.2.9 Star Tracker

The TCS subsystem shall provide an operational interface temperature of TBD to TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.10 Trim Motor

The TCS subsystem shall provide an operational interface temperature of -40 to 80 degrees C for the operational phases of the mission.

3.2.1.11.1.2.11 Trim Tab Gradients

The TCS subsystem shall provide an operational surface temperature gradient of TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.12 Release Devices

The TCS subsystem shall provide an operational interface temperature of TBD to TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.13 Array Cells

The TCS subsystem shall provide an operational interface temperature of TBD to TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.14 Array Gradients

The TCS subsystem shall provide an operational surface temperature gradient of TBD degrees C for the operational phases of the mission.

3.2.1.11.1.2.15 Enclosure Panel Gradients

The TCS subsystem shall provide an operational surface temperature gradient of TBD degrees C for the operational phases of the mission.

3.2.1.11.2 Heat Transfer

3.2.1.11.2.1 Instrument Interface Conduction

TBD

3.2.1.11.2.2 Instrument Interface Radiation

TBD

3.2.1.11.3 Optical / Thermal Properties Variations

3.2.1.11.3.1 Enclosure Panels

TBD

3.2.1.11.3.2 Electronics Deck

TBD

3.2.1.11.3.3 Solar Array Panels

TBD

3.2.1.11.3.4 Sun Shade Webs

TBD

3.2.1.11.3.5 Trim Tabs

TBD

3.2.1.11.3.6 AKM Cavity

TBD

3.2.1.11.4 Thermal Distortion

3.2.1.11.4.1 Electronics Deck

TBD

3.2.1.11.4.2 Solar Array Panels

TBD

3.2.1.11.4.3 Sun Shade Webs

TBD

3.2.1.11.4.4 Trim Tabs

TBD

3.2.1.11.4.5 Bus

TBD

3.2.1.11.5 Environments

The follow thermal environments shall be used as thermal loads in the thermal finite element model for worst case analyses.

3.2.1.11.5.1 Solar Flux

TBD

3.2.1.11.5.2 Albedo

TBD

3.2.1.11.5.3 Earth IR

TBD

3.2.1.11.5.4 Eclipse Duration

TBD

3.2.1.11.5.5 Launch Thermal Environments

TBD

3.2.1.11.6 Heater Control

TBD

3.2.1.11.7 Survival Heaters

TBD

3.2.1.11.8 Grounding

TBD

3.2.1.11.9 Electro-Static Discharge

TBD

3.2.1.11.10 Thermal Model

TBD

3.2.1.11.11 Mass Allocation

The maximum mass allocation for the TCS subsystem shall not exceed TBD lbm.

3.2.1.11.12 Power

3.2.1.11.12.1 Electronics

3.2.1.11.12.1.1 Operational

150 watts shall be assumed to be dissipated on the electronics deck during the operational phase of the mission.

3.2.1.11.12.1.2 Survival

82 watts shall be assumed to be dissipated on the electronics deck during the survival phase of the mission.

3.2.1.11.12.1.3 Launch Sequence

47.7 watts shall be assumed to be dissipated on the electronics deck during the operational phase of the mission.

3.2.1.11.12.2 Heater Circuits

3.2.1.11.12.2.1 Operational

25 watts of operational heater power will be assumed to be dissipated during the operational phase of the mission.

3.2.1.11.12.2.2 Survival

93 watts of operational heater power will be assumed to be dissipated during the operational phase of the mission.

3.2.1.11.12.2.3 Launch Sequence

watts of operational heater power will be assumed to be dissipated during the operational phase of the mission.

3.2.1.11.12.2.4 Design Margins

Design margins are built into the design since all heater circuits shall be sized for a 24 volt bus voltage. Nominal bus voltage is assumed to be 28 ± 4 volts.

3.2.1.11.13 Grounding

All MLI blankets shall be redundantly grounded to the structure with no single layer exceeding 50 Ohms to any point on the structure.

3.2.1.11.14 Thermal Closeout

The thermal closeout shall be designed such that no thermal hardware will adversely affect the operation of any mechanism.

3.2.1.11.15 Proportional Heater Control

TBD

3.2.1.11.16 Bus Voltage

TBD

3.2.1.11.17 Maximum Jitter

TBD

3.2.2 Physical Characteristics

3.2.2.1 Mass Properties

3.2.2.1.1 Launch Mass Allocation

3.2.2.1.2 Lateral Center of Mass Location

3.2.2.1.3 Axial Center of Mass Location

3.2.2.1.4 Mass Moment of Inertia Allocations

3.2.2.1.5 Mass Properties Report

3.2.2.2 Dimensions and Envelope

The dimensions of the [component name] shall not exceed TBD x TBD x TBD in. (TBD x TBD x TBD cm) w, l, h.

3.2.3 Reliability

The [component name] shall be designed and built to the highest Reliability and Quality Assurance (R&QA) program level that can be implemented within the FAME schedule and cost constraints. The [component name] shall be designed to perform as specified with a 2.5-year Probability of Success of [TBD] and with a goal of five years mission duration. Degraded modes of operation are allowable in making the reliability calculations. Non-redundant, single-string designs should be used to contain costs and single-point-failure (SPF) modes will be allowable.

3.2.4 Maintainability

No scheduled and preventive maintenance shall be required to meet the performance requirements specified herein.

3.2.5 Environmental Conditions

3.2.5.1 Ground Handling and Transportation

The FAME spacecraft bus in its approved container or packaging shall meet the requirements of this document after exposure to any combination of the following ground handling and transportation environments between NRL and the ELV launch site.

a. *Ambient Air Temperature:* The ambient temperature of the air external to the shipping container will range from -10˚C to +40˚C.

b. *Ambient Pressure:* The ambient pressure will vary between that naturally occurring at sea level and at 30,000 feet.

c. *Humidity:* Humidity within the shipping container shall be controlled such that no condensation of moisture or frost occurs on the hardware.

d. *Acceleration:* The maximum steady state acceleration shall be [2.0 g (limit)] in any direction acting separately.

e. *Vibration:* When packaged or otherwise prepared for shipment, the FAME spacecraft bus shall withstand the vibration environments specified in MIL-STD-810, [Method 514, Procedure I].

f. *Shock:* The shock levels to the structure subsystem will be controlled by design of the handling and shipping container. The packaged structure subsystem shall be designed to withstand the shock environment of MIL-STD-810, [Method 516, Procedure II].

g. *Cleanliness:* The protective container or packaging shall maintain the hardware at the cleanliness level specified in NCST-D-FM007, *FAME Contamination Control Plan.*

3.2.5.2 Storage

The FAME spacecraft bus, except for batteries and separation hardware, with protective dust wrapping about the unit(s), will meet the requirements of this document after exposure to any combination of the following storage environments defined herein for one-year storage period.

a. *Ambient Air Temperature:* The ambient air temperature will be controlled to 23˚C ±10˚C.

b. *Ambient Pressure:* The ambient pressure will vary between that naturally occurring at sea level and at 1500 feet.

c. *Humidity:* The relative humidity will be maintained at a level typical of the NRL's Building A-59 payload processing facility. Appropriate measures will be implemented at all times to prevent the formation of condensation on the FAME observatory, test equipment, or protective covers.

d. *Cleanliness:* The storage facility shall be controlled to meet the environment specified in NCST-D-FM007, *FAME Contamination Control Plan*.

3.2.5.3 Prelaunch

The prelaunch phase covers those environments that occur from arrival at the ELV launch site to launch.

a. *Ambient Air Temperature:* The ambient air temperature will be maintained from 7.2˚C to 33˚C.

b. *Ambient Pressure:* The ambient pressure will vary between that naturally occurring at sea level and at 1,500 feet.

c. *Humidity:* The relative humidity will be maintained above 30%. Appropriate measures will be implemented at all times to prevent the formation of condensation on the SS, test equipment, or protective covers.

d. *Acceleration:* The maximum steady state acceleration shall be 2.0 g (limit) in any direction acting separately.

e. *Cleanliness:* The assembly, test, and preparation area will be controlled to meet the environment specified in NCST-D-FM007, *FAME Contamination Control Plan*.

3.2.5.4 Launch and Ascent

The launch and ascent phase covers those environments that occur between terminal countdown and separation from the third stage. The FAME spacecraft bus shall meet the requirements of this document during exposure to any combination of the following environments:

a. Thermal: TBD…

b. Humidity: TBD…

c. Pressure: TBD…

d. Lateral Acceleration: TBD…

e. Axial Acceleration: TBD…

f. Random Vibration, All Axes: TBD…

g. Acoustic Levels: TBD…

h. Shock: TBD…

i. Cleanliness: TBD…

3.2.5.5 On-Orbit Operation

The orbital operations phase covers those environments that occur when the observatory reaches geosynchronous altitudes. The FAME spacecraft bus shall meet the requirements of this document during exposure to any combination of the following environments:

a. *Natural Thermal Radiation:* TBD…

b. *Pressure:* The pressure environment during orbital operation will be a hard vacuum of less than 1×10^{-5} torr.

c. *Particle Radiation:* The FAME spacecraft bus will be subjected to galactic cosmic radiation, geomagnetically trapped radiation, and solar flare particles. Any part used in the observatory shall meet the requirements of this document with a minimum total radiation dose of [1×10^{4} Rads (Si)] and a minimum linear energy transfer (LET) of [30 with a goal of 36]. Any part not meeting minimum requirements shall be identified during the PDR/CDR. A factor of two shall be added as a minimum margin after a full shielding analysis has been performed.

d. *Acceleration:* TBD…

e. *Pyrotechnic Shock:* TBD…

f. *Meteoroids and Orbital Debris:* The FAME spacecraft bus shall be designed to have a minimum probability of no penetration (PNP) for micro-meteoroid and orbital debris of [0.995] for one year.

g. *Plasma:* TBD…

h. *Solar Ultraviolet Radiation:* TBD…

3.2.6 Transportability

a. *Shipping Container:* A shipping container shall be provided for storage and transportation of the spacecraft bus and its GSE. The container shall protect the spacecraft bus and its GSE from damage and contamination during transportation and storage.

b. *Packaging and Transportation:* A packaging and transportation plan or procedure shall be developed. The plan shall address special handling requirements as applicable to the unit being delivered. Transportation of any explosive devices (i.e., ordnance) shall be in accordance with the requirements of the applicable carrier.

c. *Marking:* Marking for shipment shall be appropriate for the mode of shipment.

3.3 Design and Construction

3.3.1 Design Standards, Workmanship Requirements, and Production Techniques

a. *Standards of Manufacture:* General manufacturing requirements shall be according to established supplier practices for spaceflight equipment.

b. *Contamination Control and Cleanliness:* Cleanliness requirements shall be determined and controlled on an individual basis. NRL will prepare a Contamination Control Plan using information provided by suppliers as part of the Interface Control Working Group (ICWG) process. The plan will address a cost-effective contamination control program and the methods for implementation at each stage of the developmental, integration, test, and launch processing activities.

c. *Connectors:* Connector keying or equivalent means shall be used to prevent mismating. Connectors and connections shall have durable stripes, arrows, or other indications to show the positions of alignment pins or equivalent devices to prevent improper connection.

d. *Positive Locking Devices:* Screw-type hardware for use on-orbit shall employ positive locking. Locking compounds shall not be used. As a goal, lockwashers will not be used.

3.3.2 Materials, Processes, and Parts

Materials, processes, and parts used in the fabrication of the [component name] shall be selected to assure reliability and performance in the environments specified herein and consistent with the cost and schedule constraints of the FAME program.

3.3.2.1 Materials

a. *Materials Selection for Outgassing:* Materials shall be selected for low out-gassing characteristics. FAME should use only materials exhibiting Total Mass Loss (TML) of 1.0% or less and Volatile Condensable Material (VCM) values of 0.1% or less per NRP-1124. Any materials that fail to meet these criteria shall be identified to the NRL.

b. *Metallic Materials:* Metallic materials shall be corrosion resistant by nature or shall be corrosion inhibited by means of protective coatings. Metallic materials should be selected from Table 1 of MSFC-SPEC-522B. Any materials that fail to meet these criteria shall be identified to the NRL. Base metals intended for intermetallic contact that form galvanic couples shall be plated with those metals that reduce the potential difference or shall be suitably insulated by a nonconducting finish. Electrical bonding methods shall include provisions for corrosion protection of mating surfaces. Use of dissimilar metals shall be avoided. Magnesium alloys shall not be used for electrical bonding or grounding.

c. *Magnetic Materials:* The use of magnetic materials should be avoided whenever possible. Magnetic materials shall be used only if necessary for equipment operation. The materials used should minimize permanently induced and transient magnetic fields.

d. *Materials Flammability:* The use of flammable materials shall be avoided. The supplier shall identify to NRL any flammable materials.

e. *Toxic Products and Formulations:* The use of toxic products and formulations shall be identified to the NRL.

f. *Prohibited Materials:* Any use of the following materials shall be identified to the NRL: Teflon insulation subject to “cold-flow”; Nylon, polycarbonates, polyvinylchloride (PVC); silicone grease with zinc oxide filler; cadmium; zinc; and non-fused tin-electroplated parts.

g. *Pressurized Systems:* If pressurized vessels are used, the supplier shall provide supporting design information to allow the NRL to meet range safety hazard reporting requirements. NRL will use the guidelines of MIL-STD-1522, *Safe Design and Operation of Pressurized Missile and Space Systems,* to characterize pressurized vessels.

3.3.2.2 Processes

Selected processes shall meet the requirements of the following subparagraphs.

a. *Traceability:* A system for categorizing electronic parts into sets of homogeneous groups and tracing those parts through the fabrication, assembly, test, and delivery cycles shall be maintained.

b. *Failure Reporting and Corrective Action System:* A closed loop failure reporting and corrective action system (FRACAS) for reporting, analysis, and corrective action shall be in effect for failures occurring during the acceptance testing phase. The FRACAS shall determine whether failures are caused by design deficiencies, human error, defective parts, test equipment, or software.

c. *Soldering and Other Processes:* Soldering and other processes shall be consistent with the requirements of MIL-HDBK-1547A, Paragraph 4.4.4, *Processes and Controls*. Soldering and other processes shall be specified in process specifications that employ the guidelines of MIL-STD-1546, Paragraph 5.1, *Parts, Materials, and Processes Control Program Planning*.

3.3.2.3 Electronic Piece Parts

The selection and control procedures shall emphasize quality and reliability to meet the mission requirements, including all environmental degradation effects, and to minimize total life cycle cost for the system. The materials employed in the design shall be selected to assure maximized reliability and performance in the specified environment within the volume and weight constraints. No identical parts, such as electrical connectors, fittings, etc., shall be used where inadvertent interchange of items or interconnections could cause a malfunction.

a. *Parts Selection and Use:* Electronic parts shall be designed or selected for high reliability and long life in storage, test, and in operational use in the launch environment and during on-orbit operations in the space environment. Electronic parts and materials that have been permanently installed in an assembly and which are then removed from an assembly for any reason shall not be used in any item of spaceflight hardware. Nonstandard parts may be used where standard parts do not exist or are not available (see 3.3.2.3b for definition of “Nonstandard” and “Standard” parts).

b. *EEE Parts Program:* The intent of the EEE Parts Program is to provide the highest reliability level available within the program’s cost and schedule limitations. The radiation hardness characteristics of all parts shall be established by test, similarity, or analysis. *Standard* parts shall be selected according to the following order of preference:

(4) EEE parts shall be selected according to MIL-STD-975 with a quality level no lower than Grade 2;

(5) JANS and JANB microcircuits per MIL-M-38510 if not listed in MIL-STD-975;

(6) JANTXV, JANTX, and JANS semiconductor devices;

(7) Passive devices procured under established reliability for level of “S” and “R”;

(8) Industrial grade parts (specified for -25˚C to +85˚C operation) with additional screening.

(9) All other parts selection shall be considered nonstandard and shall be identified with explanation to the NRL.

The supplier shall procure EEE parts and perform the necessary specified screening requirements according to the supplier’s standard practices for spaceflight programs. Rescreening of JANTXV and JANTX devices is not required. A Parts Control Board is not required and requirements for a coordinated parts procurement do not apply.

3.3.2.4 Electrostatic Discharge Sensitive Parts

All electrical components utilizing electrostatic discharge sensitive parts shall provide adequate protection to preclude part failure resulting from handling, shipment, or storage situation.

3.3.3 Electromagnetic Radiaton

The [component name] shall be designed and constructed such that each item is compatible with itself and with its known test, launch, and on-orbit operational environments. The [component name] shall be designed to not be susceptible to or to radiate emission levels greater than those specified in MIL-STD-461 for space systems. The [component name] supplier shall conduct the testing for Conducted Susceptibility at the [component name] assembly level using the guidelines of MIL-STD-461 (Method CS101) and shall provide the test results to NRL. The supplier shall characterize the Radiated Susceptibility of the [component name] using the guidelines of MIL-STD-461 (Method RS103) and shall provide the test results to NRL. Final test levels of RS103 will be defined in an Interface Control Document (ICD) jointly developed by NRL and the supplier. NRL will conduct system EMI/EMC tests at the spacecraft level.

3.3.4 Nameplates and Product Marking

Components and/or equipment that are interchangeable shall be identified by part number and serial number.

3.3.5 Workmanship

Equipment shall be manufactured, processed, tested, and handled such that finished items are of sufficient quality to ensure reliable operation, safety, and service life in the operational environments.

3.3.6 Interchangeability

Assemblies, components, and parts having identical part numbers shall, where practicable, meet the requirements for an interchangeable item. Interchangeable items are defined as two or more items possessing functional and physical characteristics equivalent in performance and durability and are capable of being exchanged one for the other without alteration of the items themselves or of adjoining items except for adjustment, and without selection for fit or performance.

3.3.7 Safety

3.3.8 Human Performance/Human Engineering

3.4 Documentation

3.5 Logistics

3.5.1 Maintenance

3.5.2 Supply

3.5.3 Facilities and Facility Equipment

3.6 Personnel and Training

3.6.1 Personnel

The [component name] supplier shall allow on-site technical personnel from NRL to develop an in-depth knowledge and understanding of the [component name] via "hands-on" involvement in the development process, particularly as it relates to the development, test, and use of GSE.

3.6.2 Training

Not applicable.

3.7 Precedence

The order of precedence of the requirements specified herein is:

a. Safety;

b. Mission;

c. Design to cost;

d. Performance;

e. Quality factors; and

f. All other requirements are considered equal in order of precedence.

4. QUALIFICATION PROVISIONS

4.1 General

This section describes the analyses, tests, and inspections required for the FAME spacecraft bus verification process. Verification of design, construction, and performance of the FAME spacecraft bus will assure that the hardware and software conform to the requirements stated herein. The preferred method is test, where practical, to obtain empirical data to support verification. However, to meet program technical, schedule, and cost objectives, reuse of previously qualified flight equipment may dictate use of other verification methods (e.g., inspection, analysis, and review of design documentation). NRL will implement a quality assurance program in accordance with the requirements of NCST-D-FM005, *FAME Product Assurance Plan*, to verify compliance with specified requirements.

The analyses, tests, and inspections specified in Table 4-1 (included at the end of this section) will be conducted to verify that all requirements specified in Section 3.0 have been achieved. Test requirements shall be as stated herein with planning information documented in NCST-TP-FM001, *FAME System Integration and Test Plan*.

4.2 Verification and Verification Documentation

The requirements of Section 3.0 shall be verified by one or more of the methods detailed in the Verification Requirements Checklist (Table 4-1).

a. Similarity;

b. Analysis;

c. Inspection;

d. Validation of Records;

e. Demonstration and Measurement;

f. Simulation;

g. Review of Design Documentation; and

h. Test.

Verification will be documented using the Verification Matrix. The matrix will include a separate record for each requirement to be verified. Each record will include the requirement, traceability information, verification description, compliance data, and approval block. A detailed description of the database and associated verification process will be documented in NCST-TP-FM001, *FAME System Integration and Test Plan*. All verification documentation will be made available to inspection, test, and assessment personnel. Applicable verification drawings, specifications, and procedures will be physically located at the verification site at the time of the verification event. When each verification event is complete, the information required to validate compliance to each requirement will be entered into the verification/compliance database for completion of the Verification Matrix.

4.2.1 Verification by Similarity

Verification by similarity is a method of verification that verifies a requirement based on existing results from components and assemblies of like kind and includes a review of prior relevant hardware configurations and applications. Hardware of similar design and manufacturing process that have been qualified to equivalent or more stringent specifications may be verified by similarity.

4.2.2 Verification by Analysis

A method of verification, taking the form of the processing and accumulated results and conclusions, intended to provide proof that verification of a requirement(s) has been accomplished. The analytical results may be based on engineering study, compilation or interpretation of existing information, similarity to previously verified requirements, or derived from lower level examinations, tests, demonstrations, or analyses. Analyses will be performed as specified in Table 4-1 to verify applicable requirements of Section 3.0. The analytical methods that may be used include engineering analyses in the specified technical discipline, similarity to a previously verified requirement, review of drawings and data, use of experience, or prior testing. When an analysis is specified in Table

4-1, a detailed engineering study to verify compliance with Section 3.0 of this document will be performed and documented.

4.2.2.1 Failure Mode and Effects Analysis

As part of the design process, a Failure Mode and Effects Analysis (FMEA) shall be performed according to established supplier procedures on each component “black box” at the interface only.

4.2.2.2 Reliability Analysis

As part of the design process, a system reliability analysis shall be performed using the guidelines of MIL-HDBK-217F. A reliability mathematical model based on system/subsystem/equipment functions shall be developed and maintained. This analysis will include a failure mode and effects analysis (FMEA) (paragraph TBD) and an electrical stress analysis (paragraph TBD). As part of the reliability analysis, care shall be taken to ensure that no unidentified single point failures (SPFs) exist. Any SPF that cannot be corrected shall be documented. Special care, through electrical parts screening, component derating, and/or extra testing shall be taken to minimize risks due to SPFs. The results will be used to correct any over stressed part condition or single point failure.

4.2.2.3 Worst Case Analysis

An FAME observatory worst-case analysis will be performed by NRL in accordance with [SSD-D-IM007, Interim Control Module Worst Case Analysis] to verify that the FAME observatory will meet the performance requirements anticipated during its operating life when subjected to varying parameters such as voltage or temperature. The component supplier shall provide inputs to the WCA process.

4.2.2.4 Radiation Analysis

A radiation analysis will be performed on parts planned for use to determine the existence of any parts that are susceptible to failure at a total radiation dose level of 1×10^4 Rads (Si). Any parts so identified will be eliminated or shielded to meet radiation requirements.

4.2.3 Verification by Inspection

An element of verification consisting of investigation, without the use of special laboratory appliances or procedures, to determine compliance with requirements. Examination is nondestructive and includes (but is not limited to) visual inspection, simple physical manipulation, gauging, and measurement. Inspections will be performed as specified in Table 4-1 to verify applicable requirements of Section 3.0. These inspections are to be performed before unit qualification or acceptance testing as part of the normal quality control inspection process.

4.2.4 Validation of Records

Validation of records is a method of verification that consists of a systematic review of all relevant records to demonstrate compliance with a requirement. This method occurs as part of the hardware and software buy-off process. For requirements verified by this method, the approved buy-off package will serve to certify verification.

4.2.5 Demonstration or Measurement

A method of verification that is limited to readily observable functional operation to determine compliance with requirements. This method will not require the use of special equipment or sophisticated instrumentation.

4.2.6 Simulation

Verification by simulation is a process of verifying a requirement through the use of a representative device or system that emulates the behavior of a device or system to be verified. This method is often used when direct measurement is not possible.

4.2.7 Review of Design Documentation

Verification by the review of design documentation is a method of verification that consists of a systematic review of design documentation to determine compliance with a requirement.

4.2.8 Verification by Test

A method of verification that employs technical means, including (but not limited to) the evaluation of functional operation by use of special equipment or instrumentation, simulation techniques, and the application of established principles and procedures to determine compliance with requirements. The analysis of data derived from test is an integral part of this verification method.

Verification performed by test will be conducted according to NCST-TP-FM001, *FAME System Integration and Test Plan*. Criteria and procedures for critical parameters monitoring during test will be developed and include, as appropriate, test chamber temperature, test article temperature, pressure, test voltages and currents, test acoustic spectrum and level, test vibration spectrum and level, illumination, particle or radiation flux, instrument response and telemetry, and contamination. The FAME observatory program will reuse existing hardware to the maximum extent possible and as such will verify selected design requirements via previous qualification test data rather than retest. NCST-TP-FM001, *FAME System Integration and Test Plan*, provides a verification matrix delineating the specific verification methods to be used. The FAME observatory program will use the four types of tests specified below:

a. Functional tests (paragraph 4.2.8.1) to verify in an abbreviated fashion that the unit or system is functioning;

b. Performance tests (paragraph 4.2.8.1) to demonstrate and quantify the specified electrical and mechanical performance parameters of the unit or system;

c. Qualification (paragraph 4.2.8.2) tests to verify inherent functional performance capabilities in excess of the design requirements over the specified environment, including special interface qualification tests performed at KSC or NRL using flight equivalent units; and

d. Acceptance tests (paragraph 4.2.8.3) to gain confidence that each unit has achieved the inherent design capability verified on a sample basis.

4.2.8.1 Functional/Performance Tests

Functional or performance tests will be performed before, during, and after environmental exposures as part of the acceptance and qualification test sequences. This performance check will be made according to approved test procedures. A record will be made of all data necessary to determine complete operational and performance characteristics.

4.2.8.2 Qualification Tests

Qualification tests will be conducted to demonstrate that the design and manufacturing methods used in the construction of the FAME observatory have resulted in an item that meets the specified requirements and has suitable margins when exposed to the expected operating environments. The following qualification tests shall be performed as detailed in the Verification Requirements Checklist (Table 4-1).

4.2.8.2.1 Modal Survey

Tests may be conducted to achieve agreement between the analytical structural model and the space segment structure as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.2 Vibration

Verification will be conducted to assure compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.3 Acoustics

Verification will be conducted to assure compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.4 Pyrotechnic Shock

Verification will be conducted to assure compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.5 Thermal Vacuum

Tests will be conducted to verify compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.6 Thermal Balance

Tests will be conducted to verify the FAME observatory thermal design at the component level as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.2.7 EMI/EMC

Tests will be conducted to verify compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.3 Acceptance Tests

Acceptance tests will be conducted to demonstrate acceptability of an item for movement to the next stage of testing or buy-off. Acceptance tests are intended to act as a quality screening and process control tool to detect deficiencies of workmanship, material, and quality. The following acceptance tests shall be performed as detailed in Table 4-1.

4.2.8.3.1 Acoustics

Tests will be conducted to verify compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.3.2 Random Vibration

Tests will be conducted to verify compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.3.3 Thermal Vacuum

Tests will be conducted to verify compliance with paragraph TBD as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.8.3.4 Pyrotechnic Shock

Live firing tests will be conducted (to the maximum extent possible) to verify compliance with paragraph 3.2.8.7 as detailed in NCST-TP-FM001, System Integration and Test Plan.

4.2.9 Verification of Safety Requirements

Safety related requirements will be verified as part of the safety review process. A database of all safety hazard reports will be created that shows traceability from each of the safety requirements of this specification to each related hazard report. The database will include a definition of the verification method to be used to assure compliance for each hazard report.

4.2.9.1 Software Verification Tests

Existing test automation and requirements traceability software will be used to the maximum practical extent in support of verification activities. Test software interfacing with flight hardware will be verified prior to verification tests.

4.2.9.2 Independent Validation of Computer Programs

All computer programs performing on-line mission-critical operational functions in any FAME observatory system segment will be subject to an independent verification and validation (IV&V).

4.3 Verification Level

Verification levels are those hardware levels used to identify discrete verification activities. Verification of the FAME observatory requirements delineated in Section 3 shall be performed as specified in the Verification Requirements Checklist (Table 4-1).

4.3.1 Component

A functional unit that is viewed as an entity for the purposes of analysis, manufacturing, maintenance, or record keeping. Examples include actuators, valves, and individual “black boxes”.

4.3.2 Subsystem

A combination of items (components, structures, and interconnections) that perform a major task. Major FAME observatory subsystems are listed in paragraph 3.7.

4.3.3 System

An integrated set of all subsystems, components, and interconnections that form the complete flight item.

Table 4-1. Verification Requirements

Paragraph	Title	Verification Method											Level	Reference
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Test					
									Functional	Performance	Qualification	Acceptance		

Verification Level:
1 = Component
2 = Subsystem
3 = System

Paragraph	Title	Verification Method											Level	Reference
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Test					
									Functional	Performance	Qualification	Acceptance		

Verification Level:
1 = Component
2 = Subsystem
3 = System

Paragraph	Title	Verification Method											Level	Reference
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Test					
									Functional	Performance	Qualification	Acceptance		

Verification Level:
1 = Component
2 = Subsystem
3 = System

Paragraph	Title	Verification Method											Level	Reference
									Test					
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Functional	Performance	Qualification	Acceptance		

Verification Level:

1 = Component
2 = Subsystem
3 = System

Paragraph	Title	Verification Method											Level	Reference
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Test					
									Functional	Performance	Qualification	Acceptance		

Verification Level:
1 = Component
2 = Subsystem
3 = System

		Verification Method												
									Test					
Paragraph	Title	Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Functional	Performance	Qualification	Acceptance	Level	Reference

Verification Level:
1 = Component
2 = Subsystem
3 = System

Paragraph	Title	Verification Method											Level	Reference
									Test					
		Similarity	Analysis	Inspection	Records	Demonstration or Measurement	Simulation	Review	Functional	Performance	Qualification	Acceptance		

Verification Level:
1 = Component
2 = Subsystem
3 = System

5. PREPARATION FOR DELIVERY

5.1 Packaging and Transportation

A packaging and transportation plan or procedure shall be developed using NHB 6000.1C as a guide. Transportation of any explosive devices shall be in accordance with the requirements of the applicable carrier.

5.2 Marking

Marking for shipment shall be appropriate for the mode of shipment.

5.3 Storage

The FAME spacecraft bus, GSE, AKM, components, and spares, except for items specifically identified as being age sensitive, shall be capable of being stored for periods of at least two years without requiring major repair or maintenance at the end of storage.

6. NOTES

This section shall contain any general information that aids in understanding this document (e.g., background information, glossary, rationale). This section shall include an alphabetical listing of all acronyms, abbreviations, and their meanings and a list of any terms and definitions needed to understand this document.

6.1 Acronyms and Abbreviations